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RESEARCH MEMORANDUM

AN ANALYSIS OF DUCTED-AIRFOIL RAM

JETS FOR SUPERSONIC AIRCRAFT

By

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SUMMARY

The conventional ducted-fuselage ram jet is handicapped by inadequate storage space to house a pay load. To determine a means of overcoming this difficulty an analysis was made of a configuration in which the ram jets are housed in the wing or tail and the fuselage is devoted exclusively to the housing of cargo and controls.

Ducted-airfoil ram jets are shown to be either potentially long-range aircraft or high-acceleration interceptor-type missiles. Ranges of 1800 miles at a flight Mach number of 2.0 were calculated. Accelerations as high as 16 times gravity are obtainable with reasonable missile dimensions.

The effect of sweepback on the external aerodynamics of ducted airfoils with the leading edge ahead of the Mach cone is studied and shown to be small. The possible total pressure recoveries using two-dimensional wedge-type diffuser inlets are then determined. Using these recoveries, ducted-airfoil ram-jet thrust and propulsive coefficients are computed for hydrocarbon fuels over a wide range of fuel-air ratio. The possible range and acceleration performances are determined for aircraft with a fuselage consisting of a parabolic body of revolution with a fineness ratio of 10 and ducted airfoils of various sizes relative to the fuselage size. Both ducted-tail and ducted-wing calculations are made.

By a direct comparison of specific designs, the ducted-tail ram jet is shown to have 19 percent greater range than the ducted-fuselage of the same gross weight, or to have about two-thirds of the weight and occupy one-fifth the volume of a ducted-fuselage ram jet of the same pay load.

INTRODUCTION

In the most generally considered application of a supersonic ram jet, the ram-jet duct runs longitudinally through a body or fuselage. A

typical arrangement of recent designs, shown in figure 1(a), has an island in the diffuser air stream. The conical nose of the island, projecting forward into the air stream, forms an efficient supersonic diffuser. The main body of the island is used to house the pay load, fuel, controls, and other necessary items. The channel between the island and the outer shell is the subsonic diffuser.

Two serious and unavoidable qualities handicap this configuration: lack of storage space for pay load and lack of accessibility to this space.

The greater part of the body displacement is devoted to ducting while a relatively minor volume is devoted to storage space for useful load. Increasing the length of the island and diffuser passageway to increase the carrying capacity also increases the duct volume. Carrying this process to an extreme adds excessively to the internal duct losses and reduces the available thrust coefficient. Either the carrying capacity is small or the ram jet is bulky and cumbersome. In the latter case, the gross weight also increases and the launching problem is difficult.

In the configuration described, complicated controls and instrumentation are housed within the innerbody. Accessibility to the innerbody for making adjustments when checking the missile prior to launching presents a serious handling problem, particularly on missiles used in development work. An alternate arrangement is to eliminate the island, carrying the duct down the center of the body and housing the pay load in the annular space around the duct. This arrangement can be used on models of sufficient size to make the annular space between the duct and outer shell practical for use. However, a large percentage of wasted space allotted to ducting is not avoided.

An arrangement which overcomes both of the principal objections to the conventional configurations is obtained by housing the ram jet in the airfoils. By doing so, the entire body space can be devoted to carrying useful load. For equal load-carrying capacity the body has the same volume as the island in the diffuser of the conventional configuration and the over-all size becomes much smaller. The design becomes small and compact.

The ram-jet engine may be placed in either a ducted tail as in figure 1(b) or in a ducted wing as in figure 1(c). A Canard type with ducted airfoils aft of the center of gravity and control surfaces forward is also a logical arrangement. Typical ducted airfoil sections are shown in figures 1(d) and 1(e). The forward portion of the airfoil consists essentially of inclined planes as in a diamond airfoil section.

By separating the ram-jet function from the body function, storage capacity and accessibility are restored. However, it is to be expected that some new problems will be associated with the ducted airfoil configurations. The possibility of obtaining a very short flame-length burner

to operate at the required fuel-air ratios and combustion chamber Mach number has not been proved at the time of writing although progress has been made in this direction. New structural and diffuser problems will also be encountered. Only after these problems are surmounted can the advantages herein set forth be realized.

Many performance computations and tests have been made on the ducted-fuselage type. It remains to demonstrate the performance possibilities of the ducted-airfoil type. This paper is devoted to this task. No attempt is made to make an exhaustive survey of the various possible configurations for ducted airfoils, particularly of the possible types of diffuser. Fairly simple and readily predictable configurations have been selected to obtain reliable calculated results. In order to secure valid answers, some effort has been put into obtaining proportions which should give nearly the best results for the given type. The design studies necessary to select the types investigated and to secure good proportions of these particular types are presented, as well as the performance calculations and comparisons which are the real purpose of the paper.

ANALYSIS

A supersonic airfoil carrying a ram-jet duct is actually a supersonic biplane. The internal flow increases both the slope of the lift curve and the lift-to-drag ratio. Pressure drag should be low if the duct entrance height approaches the wing depth as is expected at high speeds or low fuel-air ratios. A practical arrangement to obtain maximum thrust and at the same time to minimize pressure drag at flight Mach numbers in the vicinity of 2.0 is to locate the combustion chamber between parallel walls with a choking outlet. In this manner the pressure drag at the after portion of the airfoil is completely removed. The choking outlet at a flight Mach number of 2.0 permits the highest rate of air flow per unit frontal area and the highest possible thrust coefficient based on projected frontal area. On the other hand, it should be noted that the high air velocity at the combustion-chamber entrance may make the burner design problem more severe than is the case when a converging-diverging nozzle is used. Although the use of a supersonic nozzle at the combustion-chamber exit (not a choking outlet) gives an increase in operating efficiency, this case is not considered for the sake of brevity.

The small chord of missile wings or tail surfaces necessitates the use of ram-jet burners with a short combustion-chamber length. While a short combustion chamber is desirable in the ducted fuselage, a short combustion chamber is a practical necessity in a ducted airfoil of small size. Most ram-jet burners at the present time require combustion chambers of from $3\frac{1}{2}$ to 6 or more feet in length. The practical use of ducted airfoils on small- and medium-size missiles requires the development of burners with a short flame length.

The mechanical and thermodynamic aspects of the combustion problem are essentially thorough mixing of the fuel and air in the desired proportions and raising the temperature to the ignition temperature. For flight at Mach numbers of three or over, where the ram temperatures of the air are above the ignition temperature of the fuel, the combustion problem is therefore greatly simplified.

Most burners at the present time operate on sprayed liquid fuel. If the use of vaporized fuel proves to be advantageous in reducing the length of combustion chambers required, it should be pointed out that the problem of generating vaporized fuel on the ram jet in flight does not appear prohibitive. Although it may be possible to accomplish vaporization with heat exchangers, it appears simpler on short-range missiles, in which tanks may be pressurized, to add the heat of vaporization to the fuel prior to launching by means of a heating element in the tank. The fuel passes through a pressure-reduction or throttle valve in the feed line and flashes to a vapor. On long-range missiles the fuel can be vaporized by preburning a very small portion of the fuel to vaporize the remainder by direct contact in a flash boiler similar to those used in submarine torpedoes. Any heat so expended is fully utilized by the ram-jet motor so that the cost is trivial.

DESIGN CONSIDERATIONS

Angle of Sweep

One of the problems confronting the designer of a ducted-airfoil ram-jet powered aircraft or missile is the choice of swept or unswept ducted airfoils. The lift and drag of ducted airfoils having a section similar to figure 1(e), with a nose slope of 3° and an aspect ratio of 2, was investigated over a range of sweepback angles. The ratio of entrance height to combustion-chamber height, 0.703, was calculated for a stoichiometric fuel-air ratio. The skin-friction drag is assumed to be 0.0030. Possible variation of the value of skin friction with angle of sweepback is not taken into account.

The airfoil duct design is such that the external flow is independent of the internal flow. It is therefore possible to calculate the pressure distribution due to shape by the method of reference 1. The lifting pressures except those due to tip effect are determined by the same method by assuming that the equivalent flat plate at angle of attack acts in a manner similar to one surface of a wedge. The tip effect on the lift is accounted for by considering that 50 percent of the tip area is effective as a two-dimensional lifting surface. That the tip effectiveness is 50 percent for the sweepback angles considered is demonstrated in appendix A.

Sweepback up to the angle of detachment of the shock wave from the lower leading edge was investigated at flight Mach numbers of 2.0 and 3.0.

Untapered airfoils appear to be suitable for housing ram jets because a constant combustion-chamber length is obtained at all spanwise stations. Untapered airfoils with an aspect ratio of 2.0 were investigated. Sweepback was obtained by shearing the airfoil back and keeping the span and the chord in the stream direction constant. The projected frontal area of the ram-jet duct is thereby kept constant. Lift-drag ratio L/D is plotted against angle of attack in figure 2(a) for a flight Mach number of 2.0 and for sweepback angles of 0° and 30° . The lift due to internal flow is taken into account. The maximum L/D , being about equal, indicates that the two airfoils are about equal aerodynamically as wing surfaces. In figure 2(b), the maximum values from a series of L/D curves, such as shown in figure 2(a), are plotted against sweepback angle. The dotted curve is computed neglecting the lift forces of the internal air flow while the solid-line curves are computed taking the lift forces of the internal air flow at unit thrust coefficient into consideration. Maximum L/D is increased by 15 percent by the internal air forces. Figure 2(c) is similar to figure 2(b) but is for a flight Mach number of 3.0. In this case the internal air flow increased maximum L/D by about 23 percent. At either a flight Mach number of 2.0 or 3.0 the change in maximum L/D over the range of angles with the leading edge ahead of the Mach line is so small that the lift-drag ratio may be disregarded as an important factor in choosing sweepback angle. The angle can be chosen from other considerations such as diffuser design. It therefore appears logical to make the first diffuser investigations for the simpler unswept case.

Optimum External Surface Angles

Diffusers in nonswept ducted airfoils having a two-dimensional wedge protruding to create oblique shock waves were investigated. In the section shown in figure 1(d), the single oblique shock wave is followed by a normal shock wave perpendicular to the wedge surface. In the diffuser shown in figure 1(e), a second oblique shock wave is reflected from the inner surface of the diffuser shell. The second oblique shock wave is followed by a normal shock wave perpendicular to the wing skin. Use of the reflected shock is the same in principle as the use of a compound wedge which generates two shock fronts. The recovery with the reflected shock wave is affected by the angle of the outer surface. If β is the half-wedge angle and δ is the angle of divergence of the diffuser shell, the entering air is turned through an angle $\beta - \delta$ by the diffuser shell.

Consider a diffuser shell with a skin-friction coefficient C_f , a projected frontal area of the sloping diffuser shell A_d , and a corresponding plan-form area S_d . The drag coefficient of the diffuser shell C_d based on projected frontal area of the sloping diffuser shell A_d is, according to linearized, two-dimensional theory

$$C_d = \left(\frac{S_d}{A_d} \right) \left(\frac{4\delta^2}{\sqrt{M_o^2 - 1}} + 2C_f \right) \quad (1)$$

The ratio of surface area to projected frontal area is

$$\frac{S_d}{A_d} = \frac{1}{2\delta} \quad (2)$$

Making this substitution gives

$$C_d = \frac{2\delta}{\sqrt{M_o^2 - 1}} + \frac{C_f}{\delta} \quad (3)$$

This expression has a minimum when

$$\delta = \sqrt{\frac{C_f}{2} \sqrt{M_o^2 - 1}} \quad (4)$$

The following table gives values of δ if $C_f = 0.0030$:

Mach number	Optimum δ	
	Radians	Degrees
2.0	0.051	2.9
3.0	.065	3.7

Increasing the value of δ decreases the length of the diffuser. The calculated values of δ were rounded off to the next higher even degree for use in the following analysis.

Diffuser Wedge Angle and Total Pressure Recovery

The selection of the diffuser wedge angle is dictated by the total pressure recovery of the internal air flow. Total-pressure ratios across nonswept ducted-airfoil diffusers are shown in figure 3 as a function of diffuser wedge included angle 2β . The ratio of total pressure at the outlet of the subsonic diffuser to free-stream total pressure includes the losses in a 90-percent-efficient subsonic diffuser, as well as the losses across the oblique shock waves and a normal shock located at the minimum cross-sectional area. The recovery ratio is given for diffuser exit Mach numbers of 0.2 and 0.3.

At a flight Mach number of 2.0 for the single-oblique-shock case the optimum wedge angle is about 32° . At this angle, the total-pressure ratio is 24 percent higher than without the wedge ($2\beta = 0$). With the reflected oblique shock, the maximum total-pressure ratio occurs at 24° , a 5.7-percent increase over the recovery for a single oblique shock wave. Figure 3(b) shows corresponding curves for a flight Mach number of 3.0. The optimum wedge angle for a single oblique shock wave is about 39° . The total pressure ratio is 75 percent higher than that without the wedge. With the reflected shock the optimum wedge angle is about the same and the recovery is 118 percent higher than without the wedge and 42.5 percent higher than with but the single oblique shock. The use of the reflected shock wave is a great advantage at a flight Mach number of 3.0. It will be noted that the total-pressure ratio falls off more rapidly at wedge angles above optimum than at those below optimum. The optimum wedge angles are well below the shock-wave detachment angles of 46° at a Mach number of 2.0 and 68° at a Mach number of 3.0.

Boundary-Layer Considerations

The low-energy air of the boundary layer would cause a serious loss of ram recovery if included in the internal air flow to the ram-jet engine. A calculation of the boundary-layer thickness on parabolic body of revolution was made to determine the size of bleed duct which would be required to bypass the low-energy air. The methods of references 2 and 3, for aerodynamically smooth bodies, were used. Figure 4 shows the boundary-layer thickness on a body 100 inches long and 10 inches in diameter, flying at sea level at a Mach number of 2.0. The favorable pressure gradient over the greater part of the length of such a body tends to keep the boundary layer small. At stations where ducted-airfoil inlets might reasonably be located, such as at 50 and 75 percent of the length, the boundary layer is approximately 0.5 and 1.0 inch thick corresponding to 5 and 10 percent of the maximum body diameter. The displacement thickness is only 0.15 and 0.30 inch. A larger body would have a relatively smaller boundary-layer thickness; a body at higher altitude, a relatively greater thickness. Some idea of the necessary size of the wing-root bleed duct can be obtained from the above figures.

Engine Performance

Engine performance is computed utilizing the optimum diffuser wedge angles. The methods of reference 4 involve a graphical solution or a solution by trial to obtain the air flow for a choking combustion chamber. A straightforward method of calculating this case was developed and is given in appendix B. The method includes the use of coefficients which makes possible an equitable distribution of the combustion-chamber losses between flameholders or fuel-distributing devices at the entrance to the combustion chamber and combustion-chamber wall friction. Thrust coefficients were worked out assuming subsonic diffuser and combustion chamber efficiencies of 90 percent, variable specific heats of the air and

combustion gases, and a burner parasite pressure loss of 1.25 times the burner entrance dynamic pressure. The assumptions and constants used are given in greater detail in appendix C. The thrust coefficients were arbitrarily reduced 15 percent to base them on the projected frontal area of the airfoil rather than on the cross-sectional area of the combustion chamber. This allows for a boundary-layer duct occupying 15 percent of the projected frontal area of the airfoil.

The resulting thrust coefficients are shown in figure 5(a) as a function of fuel-air ratio. Curves are given for sea level and 35,330-foot altitude for single and double oblique shock waves at a flight Mach number of 2.0. The values given for 35,330-foot altitude apply for all higher altitudes used in the analysis, where the temperature and composition of the atmosphere remain unchanged. The ratio of the free-stream area of the entering air to the combustion-chamber area A_0/A_4 is given in figure 5(b), and the diffuser Mach numbers and total-pressure-recovery ratios required for design computations are given in appendix C.

Ducted-Airfoil Drag Coefficients

The internal air-flow rate controls the drag coefficient through a direct effect on duct and airfoil geometry. The manner in which fuel-air-ratio effect controls the drag coefficient through its effect on air-flow rate will be investigated.

If the external diffuser surfaces have a constant slope δ , the diffuser length is a linear function of the airfoil height h and the ratio of diffuser inlet to exit area A_0/A_4 as shown in the following expression for diffuser length

$$l_d = \frac{h}{2\delta} \left(1 - \frac{A_0}{A_4} \right) \quad (5)$$

The effect of A_0/A_4 on drag coefficient will be illustrated on a ducted airfoil with a ratio of combustion-chamber height to length, $h/l_c = 0.5$ and an aspect ratio of 1.15. A value of δ of 3.0° and a surface skin-friction coefficient of 0.0030 at a flight Mach number of 2.0 result in the drag coefficients shown in figure 6(a). It will be observed that the drag coefficient also varies linearly with A_0/A_4 . In figure 6(b) is shown the manner in which the drag coefficient for the same imposed conditions varies with fuel-air ratio. The 90-percent increase in C_D over the range of fuel-air ratio shown demonstrates that fuel-air ratio has a major controlling effect on the airfoil drag coefficient.

The effect of aspect ratio and scale on drag coefficient are illustrated in figure 6(c). The drag coefficients are given for a 1-foot

combustion-chamber length while the projected frontal area is varied. The drag coefficients for aspect ratios of 1.0 and 2.0 are nearly equal. However, as the frontal area approaches zero, the chord length approaches the combustion-chamber length and the smaller aspect ratio is a little superior. The conclusion is reached that aspect ratio and scale have small effects on C_D except at very small values of frontal area relative to $(l_c)^2$. Inspection of figure 6(c) shows that the drag coefficients at fuel-air ratios of 0.067 and 0.0167 vary by a factor of about 2 to 1 over a wide range of S_f .

Propulsive Coefficients

To utilize successfully the ducted airfoil as a propulsion device the thrust must be appreciably higher than the airfoil drag. A convenient method of analysis is to define a propulsive coefficient which is equal to the difference in thrust coefficient and parasite drag coefficient of the ducted airfoil as in the equation

$$C_p = C_t - C_D \quad (6)$$

Each of the above coefficients is based on the projected frontal area of the wing.

The individual variation of C_t and C_D with fuel-air ratio was shown in figures 5(a) and 6(b). Combining these results gives the propulsive coefficients of figure 7. The parasite drag of the ducted airfoils is only of the order of 8 percent of the thrust coefficients, and the propulsive coefficients are only about 8 percent less than the thrust coefficients over the range of fuel-air ratios and for the cases considered. The trend of the thrust coefficient with fuel-air ratio governs the trend of the propulsive coefficients, the highest values occurring at the fuel-air ratio of 0.067 (the value for a stoichiometric mixture).

AIRCRAFT PERFORMANCE

Performance computations are made on aircraft having a fuselage which is a parabolic body of revolution with a fineness ratio of 10. Propulsive thrust is obtained by rectangular ducted airfoils. The propulsive coefficients are equal to those of figure 7 for the single oblique shock wave. The ratio of ducted airfoil size to body size is varied widely. For the case with the power plants in the tail surfaces the wings have 4-percent circular-arc sections, are unswept, and have raked tips giving them two-dimensional characteristics. For the case

with the power plants in the wings, swept tails were incorporated. Details regarding the drag constants used are given in appendix C. All performances are for a flight Mach number of 2.0.

Ducted Tail

Consider the tail surfaces as the ducted airfoil used for propulsive purposes. The equation of motion for accelerated level flight is obtained by equating the net propulsive thrust of the tail surfaces to the terms for body drag, wing drag, and inertia

$$C_p q_0 S_{ft} = C_{D_b} q_0 S_b + \frac{W}{(L/D)_{\max}} + \frac{W a}{g} \quad (7)$$

Rearranging the equation and substituting $q_0 = \frac{\gamma_0}{2} \rho_0 M_0^2$

$$\frac{C_p}{S_b S_{ft}} = C_{D_b} + \frac{W/S_b}{\frac{\gamma_0}{2} \rho_0 M_0^2} \left[\frac{1}{(L/D)_{\max}} + \frac{a}{g} \right] \quad (8)$$

Acceleration.— The acceleration $\frac{a}{g}$ is seen to be a function of the body load coefficient $\frac{W/S_b}{\frac{\gamma_0}{2} \rho_0 M_0^2}$ and the ratio of body frontal area to tail frontal area $\frac{S_b}{S_{ft}}$.

A solution of the above equation for a Mach number of 2.0 at sea level with C_p taken from figure 7 at fuel-air ratio of 0.067 is given as an acceleration performance chart in figure 8. The shaded areas show where bodies of various diameters lie on the chart if the total weight per unit body volume lies between 50 and 70 pounds per cubic foot. If the density is expected to be out of this range for cases of very high or low cargo density, the curves of W/S_b , which are valid for any cargo density, may be referred to. From an inspection of the obtainable accelerations, the conclusion is reached that the ducted-airfoil ram-jet powered aircraft has excellent possibilities as an interceptor-type missile. The chart is constructed for horizontal flight. For vertical flight the a/g ordinate scale should have a unit zero shift. The chart shows that small craft tend to have the highest acceleration and are logical choices for interceptor-type missiles. It further shows that for large missiles to have high acceleration the ratio S_b/S_{ft} must be small.

Economy.— To investigate the effect of fuel-air ratio on the economy of operation of a ducted-tail powered aircraft, a quantity representing the ratio of net propulsive thrust of the tail to fuel rate will be referred to as specific propulsive impulse and will be defined by the expression

$$SPI = \frac{(C_t - C_d)q_0 S_{ft}}{W_f} \quad (9)$$

where

q_0 free-stream dynamic pressure

S_{ft} projected frontal area of airfoil

W_f fuel rate, pounds per second

For a ducted-tail aircraft of a given size, operating at a given speed and altitude, it can be shown that the miles per pound of fuel for an airplane in steady flight, the increase in momentum per pound of fuel for an accelerating airplane, and the work done per pound of fuel for a climbing airplane is greatest when the ratio of propulsive thrust to fuel rate is a maximum. Utilizing the results of figure 7, the manner in which the specific propulsive impulse varies with fuel-air ratio was computed and is shown in figure 9. The maximum value occurs at fuel-air ratio of nearly 0.017 at 35,330-foot altitude or above and at about 0.25 at sea level. Lean mixtures are evidently proper for range or other operations where economy is essential. A greater specific propulsive impulse is obtained at high altitude than at sea level because the cooler atmosphere gives rise to a higher thermodynamic cycle efficiency. In this analysis the effect of varying airfoil weight with varying fuel-air ratio has been neglected. If included, this effect would probably increase the optimum fuel-air ratio.

Range.— A solution of the equation of motion with $\frac{a}{g} = 0$ gives the condition for maximum range because this gives the greatest load coefficient or fuel-load carrying capacity. A high load coefficient is probably more easily obtained by operation at high altitude and low atmospheric pressure than by operating a very large ram jet to obtain a high value of W/S_p from scale effect. The range is therefore computed for altitude operation. Range at constant fuel-air ratio and altitude is given by the equation

$$R = \frac{W_f/S_{ft}}{5280 \rho_0 g \frac{A_o}{S_{ft}} \frac{W_f}{W_a}} \quad (10)$$

where

R range, miles

w_f fuel load, pounds

W_f fuel rate, pounds per second

W_a air rate, pounds per second

A_o area of free stream entering ram duct

S_{ft} projected frontal area of ducted tail

Taking a given percentage of the weight as fuel weight (a value of 50 percent has been used) the range is proportional to the load which can be carried by the ram jet and inversely proportional to the air density. The chart of figure 10 gives range at or above an altitude of 35,330 feet in the constant-temperature zone of the atmosphere. Propulsive coefficients obtained with single oblique shock wave were used. Ranges are given for design Mach number of 2.0 at fuel-air ratios of 0.017, 0.025, 0.04, and 0.067. The greatest range is obtained with large propulsive surfaces relative to the body size. In this region an advantage is obtained by operating at less than the stoichiometric fuel-air ratio. However, the figure shows that only a limited return can be had by increasing the projected frontal area of the ducted airfoil to values greater than the body area and structural weights should be carefully checked in extreme cases to prevent an excessive lowering of the percentage of fuel weight. In the high range of values of S_b/S_{ft} and at a given value of S_b/S_{ft} , a rich mixture gives the longest range. A very lean mixture could not propel the aircraft.

Shaded areas are drawn on the diagram to show approximately where aircraft of various sizes and operating altitudes may be expected to fall on the chart. The shaded areas are drawn for total weight per unit body volume between 50 and 70 pounds per cubic foot. The range is proportional to the load coefficient $\frac{W}{S_{ft} q_o}$ and a given range is

obtainable with a large variation or interchange of altitude and diameter. The figure is labeled to indicate the range of body diameter and altitude represented by a single shaded area. For example, a range of 1800 miles is obtainable with an aircraft 32 feet in diameter at 35,330-foot altitude, and with an aircraft 1 foot in diameter at 107,000-foot altitude. Much smaller ranges are obtainable with small sizes at 35,330 feet.

If the size and operating altitude are fixed, the greatest range is obtained by increasing the size of the ducted-tail surfaces until a fuel-air ratio of about 0.017 supplies the needed thrust. This may be observed by an inspection of the shaded areas in figure 10.

Ducted Wings

Consider the ram jets to be located in the ducted wing. The equation of motion for accelerated level flight is obtained by equating the propulsive coefficient to the sum of the terms for body drag, tail drag, induced drag, and inertia. Thus

$$C_p q_o S_f = C_{D_b} q_o S_b + C_{D_t} q_o S_t + \frac{W}{L/D_1} + W \frac{a}{g} \quad (11)$$

Rearranging the equation gives

$$C_p = C_{D_b} \frac{S_b}{S_f} + C_{D_t} \frac{S_t}{S_f} + \frac{W}{S_f q_o} \left(\frac{1}{L/D_1} + \frac{a}{g} \right) \quad (12)$$

Ranges are obtained by a solution of equations (10) and (12) with $\frac{a}{g} = 0$. The ranges are computed for an altitude of 35,330 feet. The range results are shown in figure 11 for the case with the combustion-chamber height equal to half its length and for fuel-air ratios of 0.025 and 0.017. The maximum ranges obtained are only 60 percent of those obtained for the ducted tail as shown in figure 10. The reason for the reduction of maximum obtainable ranges is as follows. The load-carrying capacity as represented by the load coefficient $\frac{W}{S_f q_o}$ is nearly equal to 3 at very low values of S_b/S_f and a fuel-air ratio of 0.025. This value corresponds to a lift coefficient of nearly unity. The required angle of attack equals 12.8° and results in a high drag to lift ratio.

To obtain a substantial increase in maximum range ratio of wing plan form to combustion-chamber area must be increased or, what amounts to the same thing, the ratio of wing chord to combustion-chamber height, or fineness ratio, must be increased. The effect on range of arbitrary changes in the proportions to obtain a variation in fineness ratio is shown in figure 12 for a fuel-air ratio of 0.017. The limiting case with the ratio of body to wing frontal area equal to zero is shown, the wing aspect ratio and flight conditions are unchanged. Ranges comparable with the ducted tail are obtainable with a wing having a section fineness ratio of 16.

Comparison of Ducted Fuselage with

Ducted-Tail Configurations

Range comparisons have been made between a ducted-fuselage ram jet 2 feet in diameter and a family of ducted-tail ram jets having parabolic bodies, with a fineness ratio 10. The sizes of the ducted-tail ram jets are such that the following quantities are equal to those of the ducted fuselage:

- (a) Pay load
- (b) Weight
- (c) Diameter
- (d) Volume

The configurations compared are those of figures 1(a) and 1(b). The cargo density is 45 pounds per cubic foot, three-fourths of which is assumed to be fuel and one-fourth pay load. The comparison is made at an altitude of 35,330 feet and a flight Mach number of 2.0. The wings are of the proper size in each aircraft to operate at maximum lift-to-drag ratio. The fuel-air ratio of the ducted-tail ram jet is 0.0167, the design tail area being adjusted for the required thrust. The fuel-air ratio of the ducted fuselage required to give a thrust equal to the drag is 0.00625. Weight estimates are made to obtain the gross weights. The type of mechanical construction to obtain the breakdowns is given in appendix D. The essential features of the comparison between ducted-fuselage and ducted-tail ram jets are given in the following table:

Type	Body volume (cu ft)	Diameter (ft)	Weight (lb)	Pay load (lb)	Range (miles)
Ducted fuselage	41.8	2.0	690	96	440
Ducted tail	$\left\{ \begin{array}{l} 8.57 \\ 12.8 \\ 33.5 \\ 41.8 \end{array} \right.$	$\left\{ \begin{array}{l} 1.27 \\ 1.45 \\ 2.0 \\ 2.15 \end{array} \right.$	$\left\{ \begin{array}{l} 465 \\ 690 \\ 1750 \\ 2150 \end{array} \right.$	$\left\{ \begin{array}{l} 96 \\ 144 \\ 377 \\ 470 \end{array} \right.$	$\left\{ \begin{array}{l} 473 \\ 524 \\ 680 \\ 722 \end{array} \right.$

For equal pay load (equal cargo and fuel space) the ranges are of comparable magnitude, being $7\frac{1}{2}$ percent greater for the ducted tail. In this case the ducted-tail ram jet is a much smaller aircraft occupying only one-fifth as much volume and having two-thirds as much weight. If the volumes compared are those of cylinders circumscribing the bodies, the volume ratio becomes one-third. When the ducted-tail ram jet is made large enough to have the same body volume as the ducted fuselage, a pay load nearly five times as great is carried 64 percent farther.

The two cases are not strictly comparable because a 6-foot combustion chamber was assumed for the ducted fuselage and a 1-foot combustion chamber for the ducted airfoil. A reduction of the tail-pipe length of the ducted fuselage to 1 foot, together with moving the tail surfaces forward 5 feet, results in much larger tail surfaces and no appreciable reduction in drag. An investigation of the proper configuration and the performance of a ducted fuselage with a short combustion-chamber length is beyond the scope of this paper.

It is evident for the configurations and sizes compared that the ducted-tail ram jet has marked practical advantages over the ducted-fuselage ram jet. This is particularly true for operations such as shipboard operations where storage and launching space is at a premium. Sufficient variations of the configurations have not been made for generalization of the results, but a definite indication has been given.

CONCLUSIONS

If efficient, short-flame-length burners are available, it follows that:

1. Ducted-airfoil drag is small, being of the order of 10 percent of the obtainable thrust coefficients.
2. Ducted-airfoil ram jets show excellent promise as interceptor-type missiles. Calculations show designs of reasonable proportions to have accelerations as high as 16 times gravity at their starting weight.
3. A ducted-tail ram jet which was designed for the same pay load and fuel capacity as a ducted-fuselage ram jet had only two-thirds of the weight and one-third of the bulk of the ducted-fuselage ram jet. The ranges in this case were of comparable magnitude. When compared on the basis of equal volume, the load capacity and range of a ducted-tail ram jet may substantially exceed the corresponding values for the ducted-fuselage ram jet. In the present example, nearly five times as great a pay load is carried 64 percent farther.
4. Gains in total pressure recovery ratio by the use of a wedge and single oblique shock wave are of the order of 25 and 75 percent at flight Mach numbers of 2.0 and 3.0. Additional gains are obtainable by a reflected shock wave.
5. A lean fuel-air mixture gives the most economical operation of a ducted-airfoil ram jet if a combustion efficiency can be obtained which is equal to that of a rich mixture.
6. Except for possible trouble with combustion at low pressures, dynamic considerations show that operation at altitudes of the order of 100,000 feet is feasible with small ducted-airfoil ram jets.

RECOMMENDATIONS

1. Since the attainment of the computed performance of ducted-airfoil ram jets, particularly in the smaller sizes, is dependent on obtaining an efficient burner with a short flame length, it is recommended that every effort be made to develop such a burner.

2. Because the development of the boundary layer in the diffuser may give rise to difficulties in obtaining a high total pressure recovery ratio, steps should be taken to develop designs which minimize such difficulties.

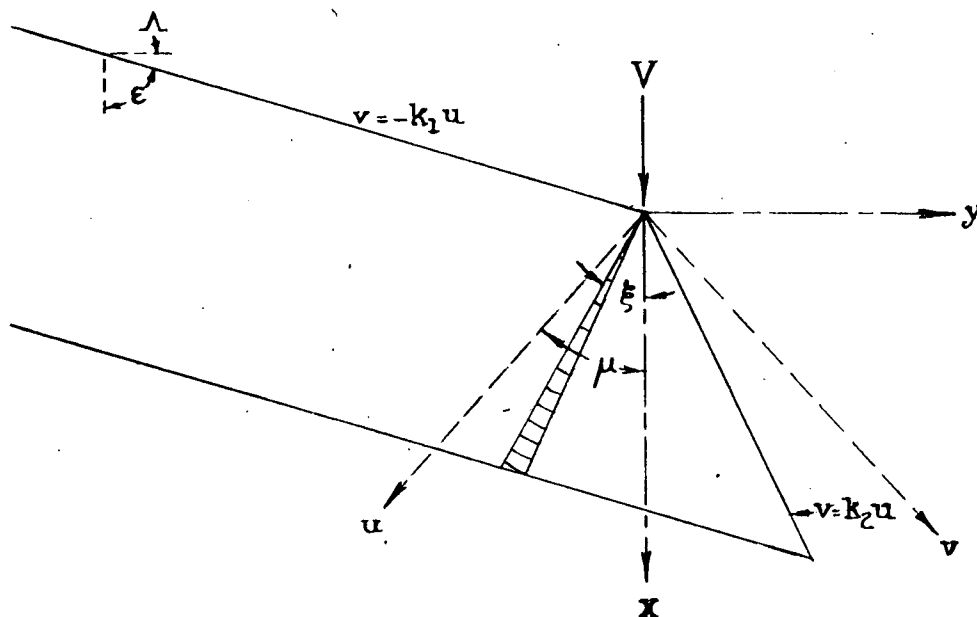
Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

APPENDIX A

TIP EFFECTIVENESS

To determine the effect of the tip on the lift of a sweptback, untapered wing, leading edge ahead of Mach line, the pressure distribution due to the tip was calculated by linearized flow theory. The disturbance-velocity potential ϕ produced by the tip is given in reference 8 as

$$\phi = \frac{V\alpha}{\pi\beta} \left\{ \sqrt{\frac{(k_1 + k_2)(x + \beta y) [(k_2 - 1)x - (k_2 + 1)\beta y]}{k_2^2}} + \frac{(1 + k_1)x + (1 - k_1)\beta y}{\sqrt{k_1}} \tan^{-1} \sqrt{\frac{k_1 [(k_2 - 1)x - (k_2 + 1)\beta y]}{(k_1 + k_2)(x + \beta y)}} \right\}$$



where

V free-stream velocity

M free-stream Mach number

u, v axes along Mach lines

α angle of attack

$$\beta = \sqrt{M^2 - 1}$$

For the constant-chord sweptback wings under consideration, tip cut off parallel to free stream

$$k_1 = \frac{\beta \tan \epsilon + 1}{\beta \tan \epsilon - 1}$$

$$k_2 = 1$$

After differentiating the potential function with respect to x to determine the disturbance velocity, substituting in the equation for pressure coefficient

$$C_p = -\frac{2}{V} \frac{\partial \phi}{\partial x}$$

and dividing by the two-dimensional pressure coefficient

$$C_{p_\infty} = -\frac{2\alpha}{\sqrt{\beta^2 - \frac{1}{\tan^2 \epsilon}}}$$

given in reference 9, the resulting equation is

$$\frac{C_p}{C_{p_\infty}} = \frac{2}{\pi} \sin^{-1} \sqrt{\frac{1 + \frac{\tan \mu}{\tan \epsilon}}{\frac{1}{\tan \epsilon} - \frac{\tan \mu}{\tan \epsilon}}}$$

To integrate the pressure over the tip, a polar incremental area dA given by the equation

$$dA = \frac{c^2}{2} \frac{\sin^2 \epsilon}{\sin^2(\xi - \epsilon)} d\xi$$

is used; where c is the chord in the free-stream direction.

The tip effectiveness, defined as the fractional part of the tip area that is equivalent to a two-dimensional lifting surface, is given by

$$\frac{\int C_p dA}{C_{p_\infty} \int dA}$$

where

$$\frac{\int C_p dA}{C_{p_\infty} \int dA} = \frac{2}{\pi} \sin^2 \epsilon (\cot \epsilon + \cot \mu) \int_{-\mu}^0 \sin^{-1} \sqrt{-\frac{1 + \frac{\tan \mu}{\tan \epsilon}}{\frac{1}{\tan \xi} - \frac{\tan \mu}{\tan \epsilon}}} \frac{d\xi}{\sin^2(\xi - \epsilon)}$$

Substituting $n = \frac{\tan \mu}{\tan \epsilon}$ and $\sigma = \frac{\tan \xi}{\tan \mu}$

the above equation reduces to

$$\frac{\int C_p dA}{C_{p_\infty} \int dA} = \frac{2}{\pi} (n + 1) \int_{-1}^0 \sin^{-1} \sqrt{-\frac{(1 + n)\sigma}{1 - n\sigma}} \frac{d\sigma}{(n\sigma - 1)^2}$$

or

$$\frac{\int C_p dA}{C_{p_\infty} \int dA} = \frac{1}{2}$$

It is therefore seen that the value of tip effectiveness of untapered wings is independent of sweepback and has a value of $\frac{1}{2}$.

APPENDIX B

INTERNAL-FLOW EQUATIONS

Symbols

a	coefficient of x^2 in quadratic equation $ax^2 + bx + c = 0$
a/g	ratio of acceleration to gravity
A	cross-sectional area of duct or air stream
b	coefficient of x in quadratic equation
c	constant in quadratic equation
C	combustion-chamber parasite or friction loss coefficient
C_D	drag coefficient, based on projected frontal area of ducted airfoil
C_t	thrust coefficient, based on projected frontal area of ducted airfoil
C_p	propulsive coefficient, $C_t - C_D$, based on projected frontal area of ducted airfoil
D	drag
d	mathematical group
e	mathematical group
F	ram-jet thrust, lb
g	acceleration of gravity (32.17 ft/sec ²)
h	airfoil height, ft
H	stagnation pressure, lb/sq ft
H.V.	lower heating value of fuel, Btu/lb
J	mechanical equivalent of heat (778 ft-lb/Btu)
l_c	combustion-chamber length
l_d	diffuser length
L	lift
M	Mach number

p	static pressure, lb/sq ft
q	dynamic pressure, lb/sq ft
q _c	impact pressure, lb/sq ft ($H - p$)
R	range, miles
R _a	gas constant for air (53.3 ft-lb/(°F)(lb))
R _m	gas constant for products of combustion, ft-lb/(°F)(lb)
S	wing area, sq ft
S _f	projected frontal area of ducted wing, sq ft
S _{ft}	projected frontal area of ducted tail sq ft
SPI	specific propulsive impulse, lb-sec/lb
T	temperature, °F absolute
V	air or gas speed, fps
w _f	weight of fuel, lb
W	gross weight of ram-jet airplane, lb
W _a	weight rate of air flow, lb/sec
W _f	weight rate of fuel flow, lb/sec
β	half angle of diffuser wedge
γ	ratio of specific heat at constant pressure to specific heat at constant volume
$\bar{\gamma}$	average value of γ between 0° F absolute and temperature T
δ	angle of diffuser outer surface with axis of airfoil
η _c	combustion efficiency
η _d	diffuser efficiency
ρ	mass density, slugs/cu ft

Subscripts:

b	body
d	diffuser
f	frontal area or friction

i induced

t tail

Subscripts denoting stations along power-plant duct:

o free stream

1 behind first oblique shock wave

2 behind second oblique shock wave

3 behind normal shock wave

4 end of diffuser; entrance to combustion chamber

5 end of combustion chamber

The method of computation of thrust coefficients and duct dimensions for a ram jet with either choking or subsonic flow at the combustion-chamber exit is developed below. For cases with the flow defined by fixing the entrance area or the Mach number at the entrance to the combustion chamber the methods of reference 4 were used. The following stations are used in the analysis of the internal flow. Station 0 refers to the free stream; the supersonic diffuser extends from station 0 to 3, the subsonic diffuser from station 3 to 4, and the combustion chamber from station 4 to 5. These stations are shown in figure 1(e).

The equations governing the flow through a constant-area combustion chamber are written below. The energy equation is

$$\left[\frac{2\bar{\gamma}_5}{(\bar{\gamma}_5 - 1)\gamma_5} + M_5^2 \right] v_{s5}^2 = \left[\frac{2\bar{\gamma}_4}{(\bar{\gamma}_4 - 1)\gamma_4} + M_4^2 \right] v_{s4}^2 + 2gJ(H.V.)\eta_c \frac{W_f}{W_a + W_f} \quad (13)$$

the momentum equation is

$$p_5 + \left(\rho_4 v_4 + \frac{W_f}{gA_4} \right) v_5 + C_4 q_4 + C_{4-5} \frac{(q_4 + q_5)}{2} - p_4 - \rho_4 v_4^2 = 0 \quad (14)$$

Here C_4 is a coefficient of pressure loss for flame holders or spray bars in the entrance to the combustion chamber and C_{4-5} is a coefficient of friction pressure loss through the combustion chamber. The continuity equation is

$$\rho_5 v_5 = \rho_4 v_4 + \frac{W_f}{gA_4} \quad (15)$$

The solution of equations (13), (14), and (15) is

$$M_4^2 = \frac{-b_4 + \sqrt{b_4^2 - 4a_4c_4}}{2a_4} \quad (16)$$

where

$$\begin{aligned}
 a_4 &= v_{s5}^2 \frac{e_4^2}{d_4} - M_5^2 \left(1 - \frac{c_4}{2} - \frac{c_{4-5}}{4} \right)^2 \\
 b_4 &= v_{s5}^2 \frac{e_4^2}{d_4} \left[\frac{2\bar{\gamma}_4}{(\bar{\gamma}_4 - 1)\gamma_4} \right] - \frac{2}{\gamma_4} M_5^2 \left(1 - \frac{c_4}{2} - \frac{c_{4-5}}{4} \right) \\
 c_4 &= -\frac{M_5^2}{\gamma_4^2} \\
 d_4 &= v_{s0}^2 \left[\frac{2\bar{\gamma}_0}{(\bar{\gamma}_0 - 1)\gamma_0} + M_0^2 \right] \left(\frac{W_a}{W_a + W_f} \right)^2 \\
 e_4 &= \frac{1}{\gamma_5} + M_5^2 \left(1 + \frac{c_{4-5}}{4} \right)
 \end{aligned}$$

The combustion-chamber exit velocity can be obtained by the energy relation

$$v_{s5}^2 = \frac{\left[\frac{2\bar{\gamma}_0}{(\bar{\gamma}_0 - 1)\gamma_0} + M_0^2 \right] v_{s0}^2 + 2gJ(HV)\eta_c \frac{W_f}{W_a + W_f}}{\frac{2\bar{\gamma}_5}{(\bar{\gamma}_5 - 1)\gamma_5} + M_5^2} \quad (17)$$

The temperature of the leaving gases is

$$T_5 = \frac{v_{s5}^2}{\gamma_5 g R_m} \quad (18)$$

The total-pressure ratio and Mach number ratio across the oblique and normal shock waves of the supersonic diffuser are given in appendix C. Taking stations 3 and 4 as the entrance and exit of the subsonic diffuser, the subsonic diffuser efficiency was

$$\eta_d = 1 - \frac{H_3 - H_4}{q_{c3} - q_{c4}} \quad (19)$$

The total pressure ratio across the subsonic diffuser is then given by the equation

$$\frac{H_4}{H_3} = \frac{\frac{H_4}{p_4}}{\frac{H_3}{p_3}} \frac{1 + \eta_d \left(\frac{H_3}{p_3} - 1 \right)}{1 + \eta_d \left(\frac{H_4}{p_4} - 1 \right)} \quad (20)$$

The temperature at the diffuser exit is a function of M_0 and the local Mach number M_4 .

$$T_4 = T_0 \frac{\frac{2\bar{\gamma}_0}{(\bar{\gamma}_0 - 1)} + \gamma_0 M_0^2}{\frac{2\bar{\gamma}_4}{(\bar{\gamma}_4 - 1)} + \gamma_4 M_4^2} \quad (21)$$

The pressure, density, and velocity at the diffuser exit are given by

$$p_4 = \frac{H_4}{\left[1 + \frac{\gamma_4 - 1}{2} M_4^2 \right] \frac{\gamma_4}{\gamma_4 - 1}} \quad (22)$$

$$\rho_4 = \frac{p_4}{gR_a T_4} \quad (23)$$

$$v_4 = M_4 \sqrt{\gamma_4 gR_a T_4} \quad (24)$$

The pressure at the combustion-chamber exit may now be obtained by equation (14).

The ratio of entrance area to combustion-chamber area is

$$\frac{A_0}{A_4} = \frac{\rho_4 v_4}{\rho_0 v_0} \quad (25)$$

The thrust per square foot of combustion-chamber cross-sectional area is

$$\frac{F}{A_4} = \rho_4 v_4 (v_5 - v_0) + \frac{\rho_4 v_4 v_5}{W_a/W_f} + (p_5 - p_0) \quad (26)$$

if no supersonic nozzle is used. If one is used, then v_5 and p_5 must be replaced by the velocity and pressure at the exit of the nozzle and the pressure term be multiplied by the ratio of exit area to combustion-chamber area. The thrust coefficient based on the projected frontal area of the wing S_f is

$$C_T = \frac{2(F/A_4)(A_4/S_f)}{\rho_0 v_0^2} \quad (27)$$

The value A_4/S_f is taken as 0.85.

APPENDIX C

CONDITIONS USED IN ANALYSIS

The conditions used in the analysis are listed in detail below:

1. The supersonic diffuser performance is calculated using the generally accepted equations for oblique and normal shock waves, reference 5. For this purpose, the value of the ratio of specific heats of the entering air was taken as 1.40. The total-pressure ratio across the supersonic diffuser and the Mach number at the entrance to the subsonic diffuser for a flight Mach number of 2.0 are given in the following table:

	Optimum wedge angle 2β (deg)	H/H_0	M
Single oblique shock	32	0.903	0.740
Double oblique shock	24	.954	.817

2. The subsonic diffuser total pressure recovery was assumed to equal 90 percent of the reduction in impact pressure.

3. The ratio of specific heats for air was varied with temperature. The ratio of specific heats for the products of combustion was varied with both temperature and the fuel-air ratio. The values of the ratio of specific heats were taken from figures 15(a) and 15(b) of reference 4. The values of the gas constant for the products of combustion were taken from figure 15(c) of the same reference.

4. The fuel is added at the entrance of the combustion chamber. The momentum required to bring the fuel up to combustion-chamber velocity is charged as a momentum pressure loss.

5. The combustion-chamber parasite losses are assumed to equal 1.25 times the entrance dynamic pressure. The lower heating value of the fuel is 19,000 Btu per pound. Ninety percent of this available energy is assumed to be liberated and appears in the products of combustion at the combustion-chamber exit. One-dimensional flow is assumed.

6. The combustion chamber is a duct with constant cross-sectional area.

7. The width of the boundary-layer bleed ducts is taken to be 15 percent of the exposed semispan. This is believed to be a conservative figure and makes some allowance for misalignment of the fuselage with the free air stream.

8. The density and temperature of the atmosphere were taken as NACA standard to an altitude of 65,000 feet and were taken from reference 7 above this altitude.

9. Nonducted wings have 4-percent circular-arc sections in the free-stream direction. The wings are not swept and have cut-forward, pointed tips which give them two-dimensional characteristics.

10. Nonducted tails have a drag coefficient of 0.04 based on body frontal area.

11. The fuselage consists of a parabolic body of revolution with the vertex at the 50-percent station and a fineness ratio of 10. The pressure drag coefficients used for the body are calculated using reference 2.

12. The skin-friction drag coefficient for airfoil (both wing and tail) surfaces is taken as 0.0030 for a single surface. Because of the higher Reynolds numbers of the flow over the body and after a study of reference 6 the skin-friction drag of the body was taken as 0.0020.

13. In calculating aircraft performance the interference of body on airfoils and airfoils on body was neglected.

14. Pressure distributions on ducted airfoils were made with linearized theory according to the methods of references 1, 8, and 9. Only airfoils with the leading edge ahead of the Mach cone were considered because of the limitations of available theory for lifting airfoils.

APPENDIX D

WEIGHT ESTIMATE

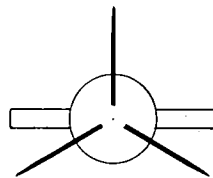
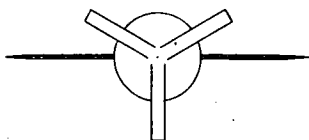
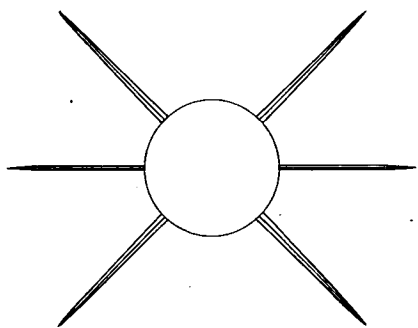
Weight calculations for the ducted-tail configurations were based on the following construction. Two dimensions are given representing the values for the smallest and greatest ducted-tail ram jets. The body and wing (4-percent circular arc) are of $\frac{1}{16}$ - to $\frac{3}{32}$ -inch magnesium, and the ducted tail of $\frac{1}{16}$ - to $\frac{3}{32}$ -inch inconel.

The construction of the ducted-fuselage configuration consists of a $\frac{1}{16}$ -inch magnesium for the inner body, tail fins, and 4-percent circular-arc wings, $\frac{3}{32}$ -inch magnesium for the diffuser, and $\frac{1}{16}$ -inch inconel for the combustion chamber.

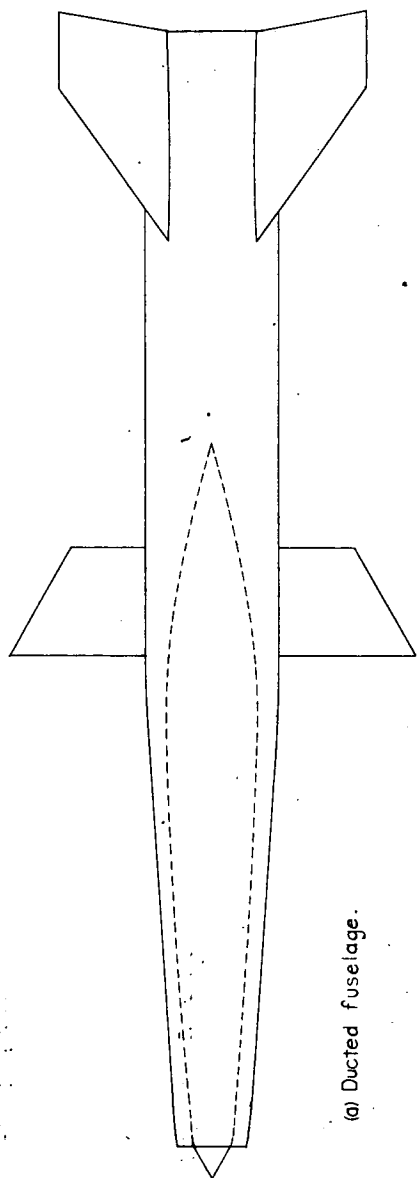
The load-carrying weight for both configurations was determined by assuming a load density of 45 pounds per cubic foot, three-fourths of which is assumed to be fuel and one-fourth pay load. Total weight for both types of ram jets includes weight of burners and needed structure. The difficulty of constructing the ducted tail from flat sheets with internal pressures has been accounted for by weight allowances for suitable stiffening material in the form of ribs and struts.

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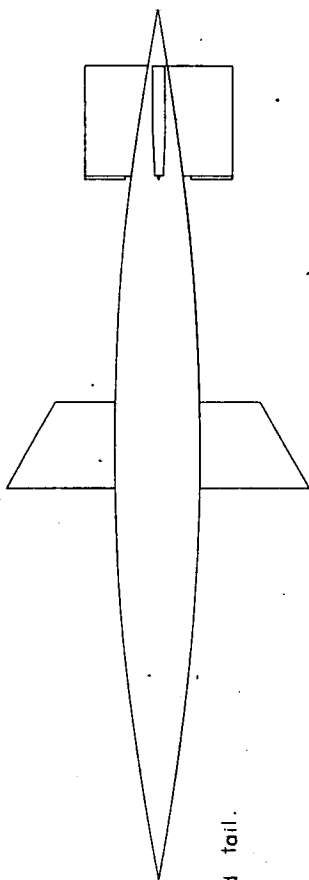
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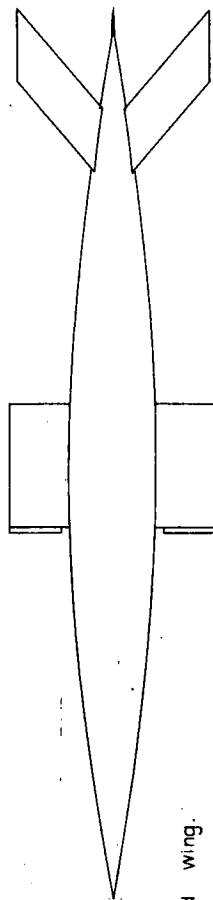
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(a) Ducted fuselage.

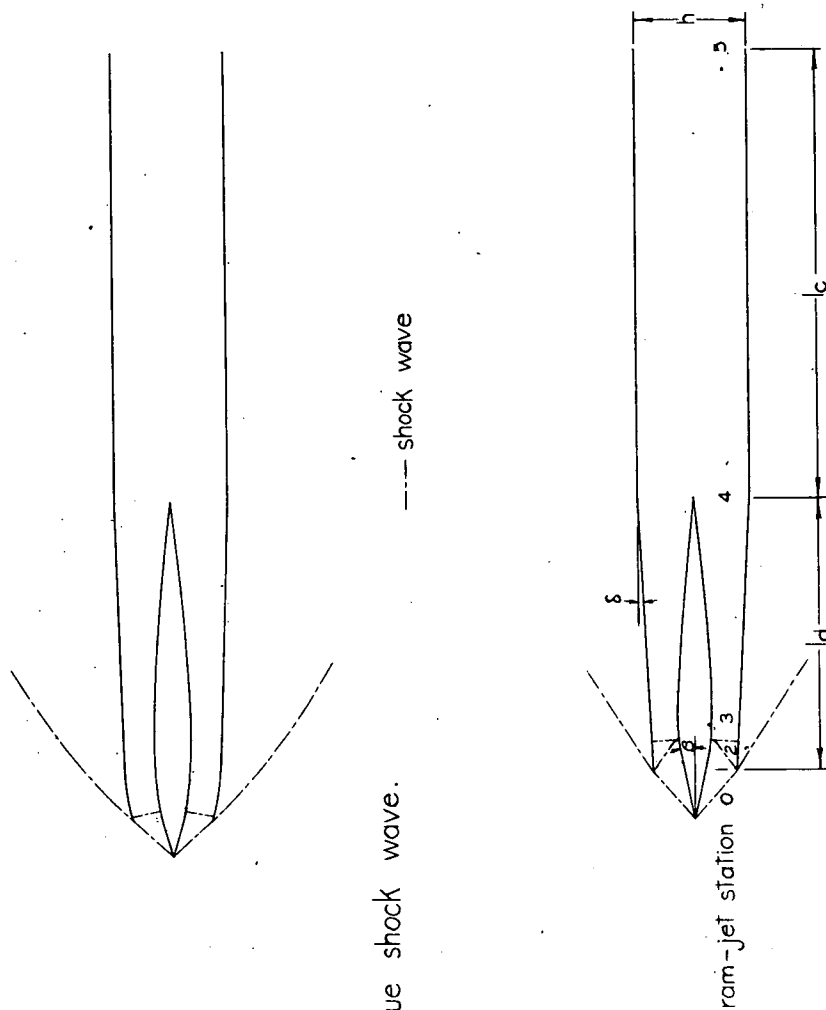


(b) Ducted tail.



(c) Ducted wing.

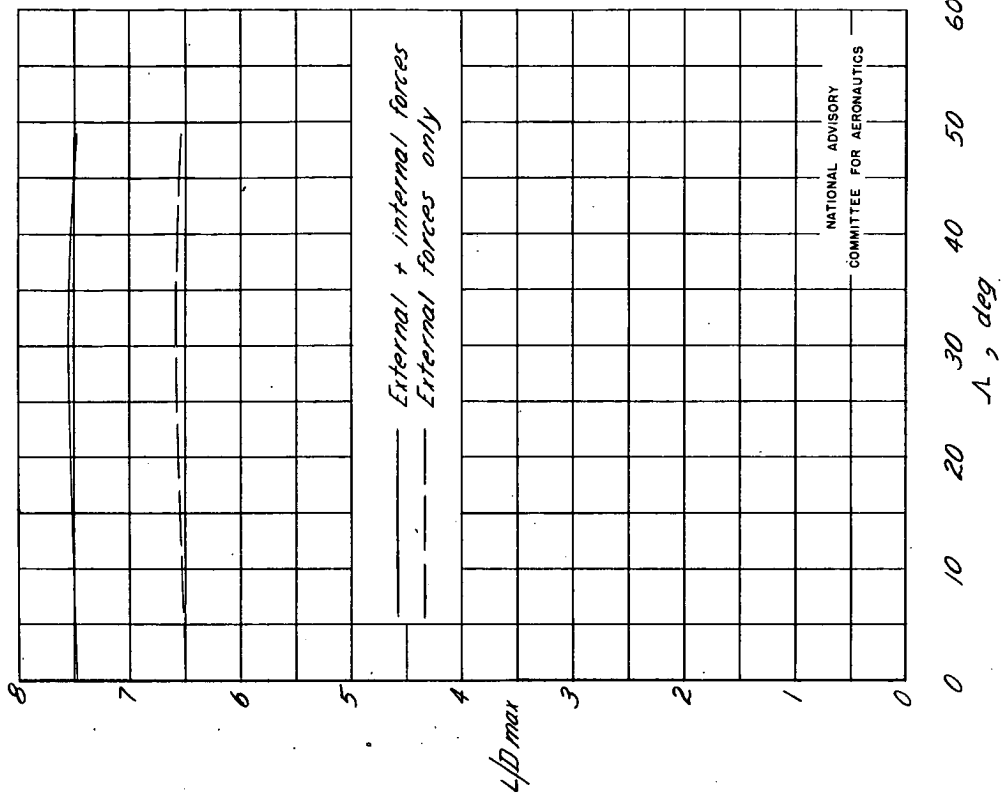
Figure 1.- Ram-jet configuration types.



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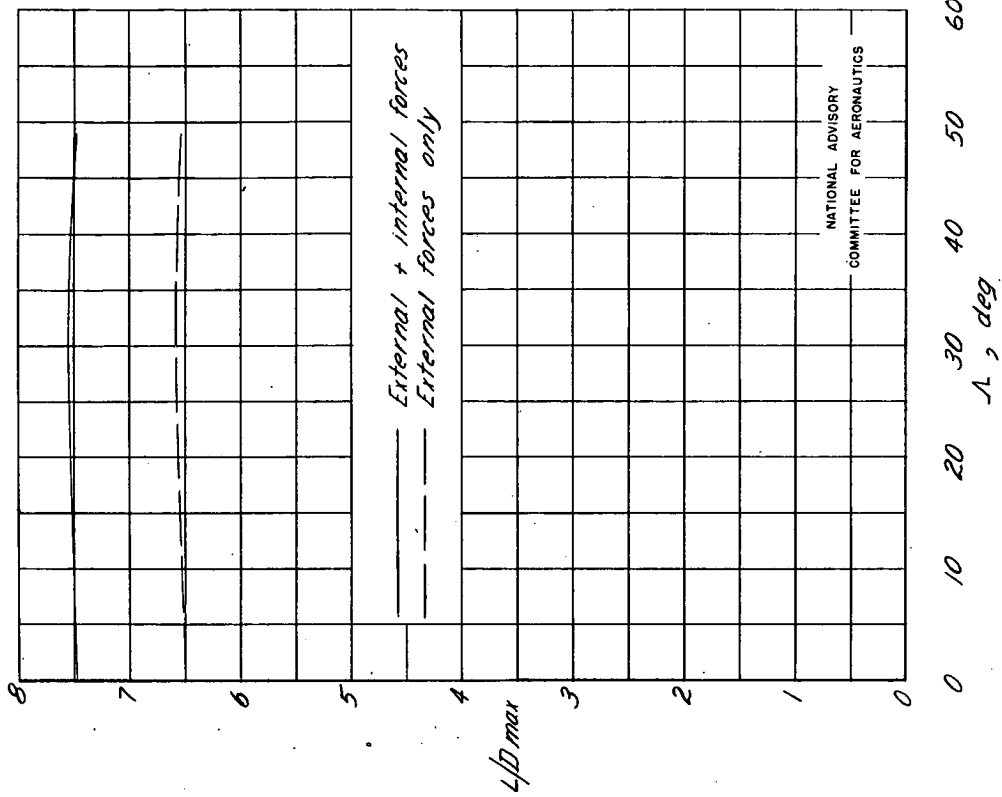
(d) Ducted airfoil section; single oblique shock wave.
(e) Ducted airfoil section; double oblique shock wave.

Figure 1.-Concluded.



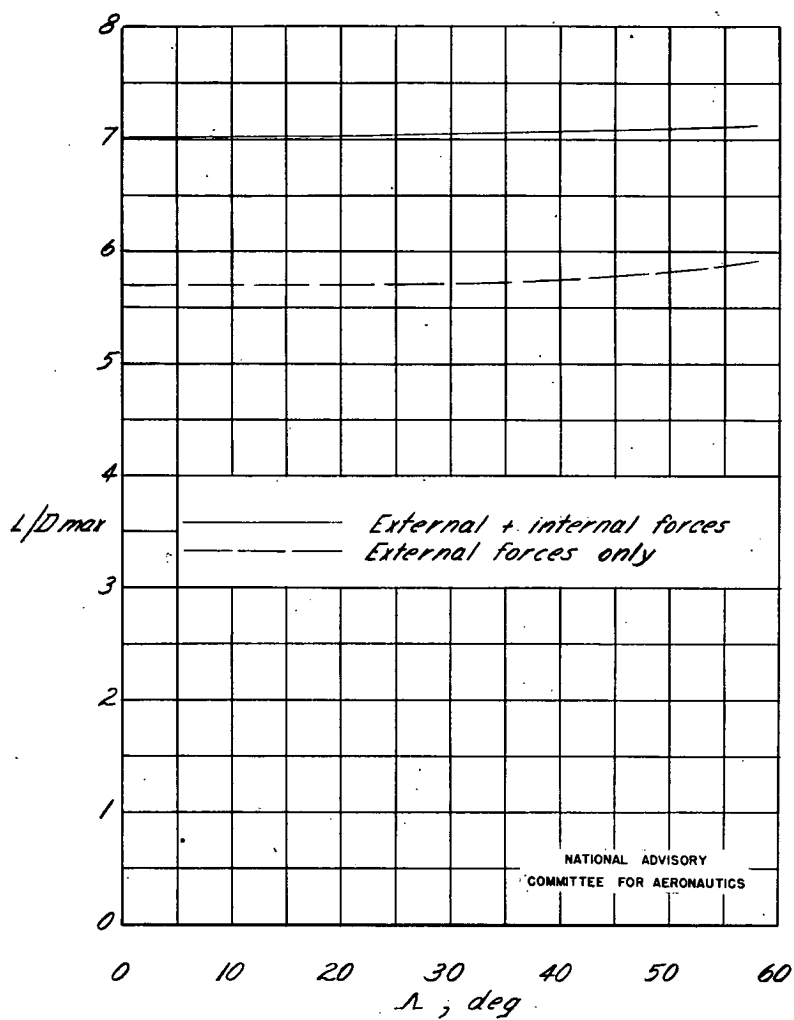
(a) Effect of angle of attack ; $M=2.0$.

Figure 2.- Lift-to-drag ratio. Aspect ratio, 2.0. Internal lift included.



(b) Effect of sweepback angle ; $M=2.0$.

Figure 2. - Continued.



(c) Effect of sweepback angle ; $M = 3.0$.

Figure 2. - Concluded.

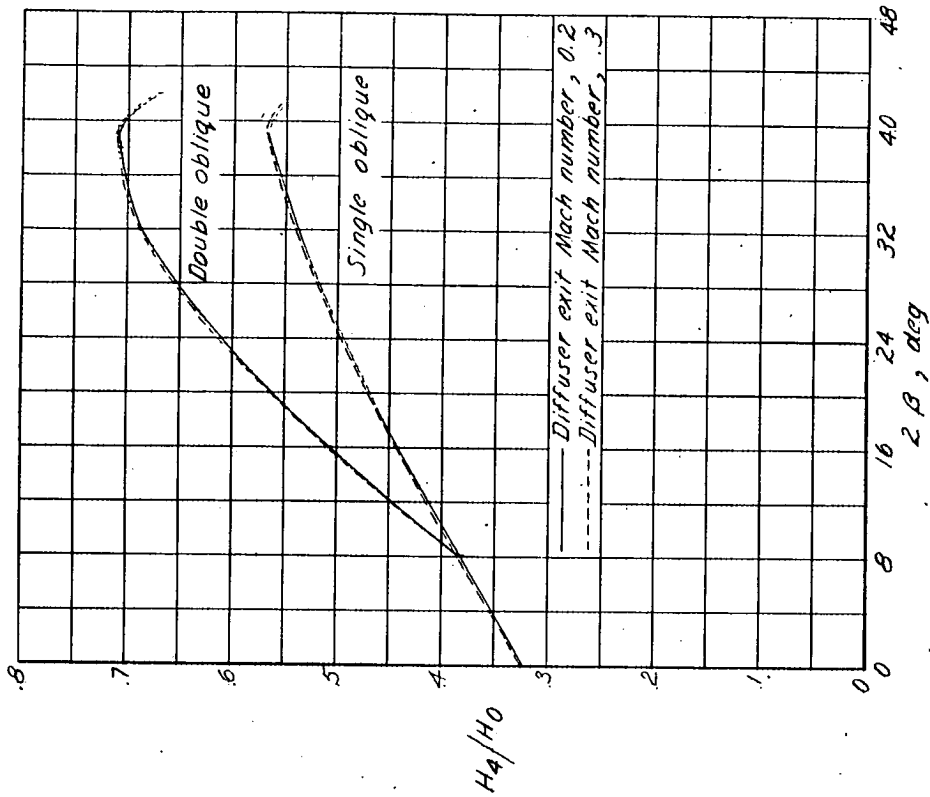
(b) $M = 3.0$ NATIONAL ADVISORY
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Figure 3. - Concluded.

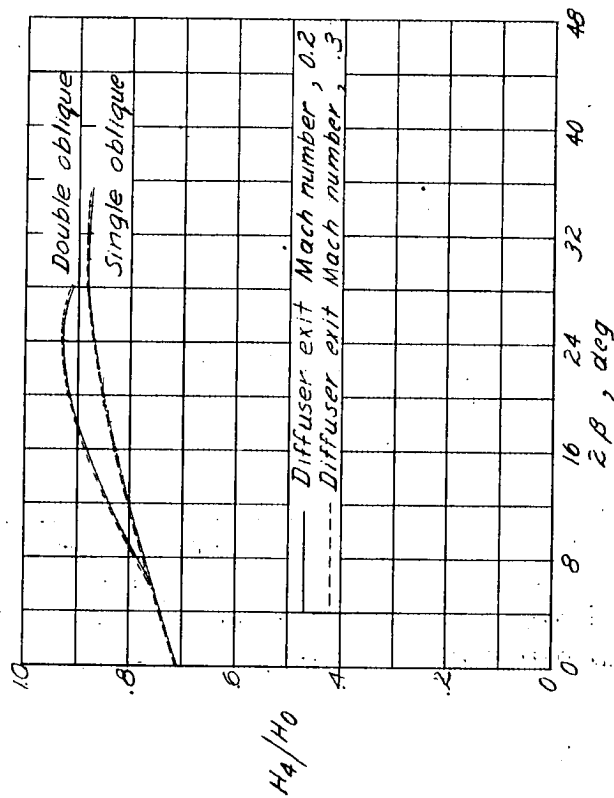
(a) $M = 2.0$

Figure 3. - Effect of diffuser wedge angle on total pressure recovery.

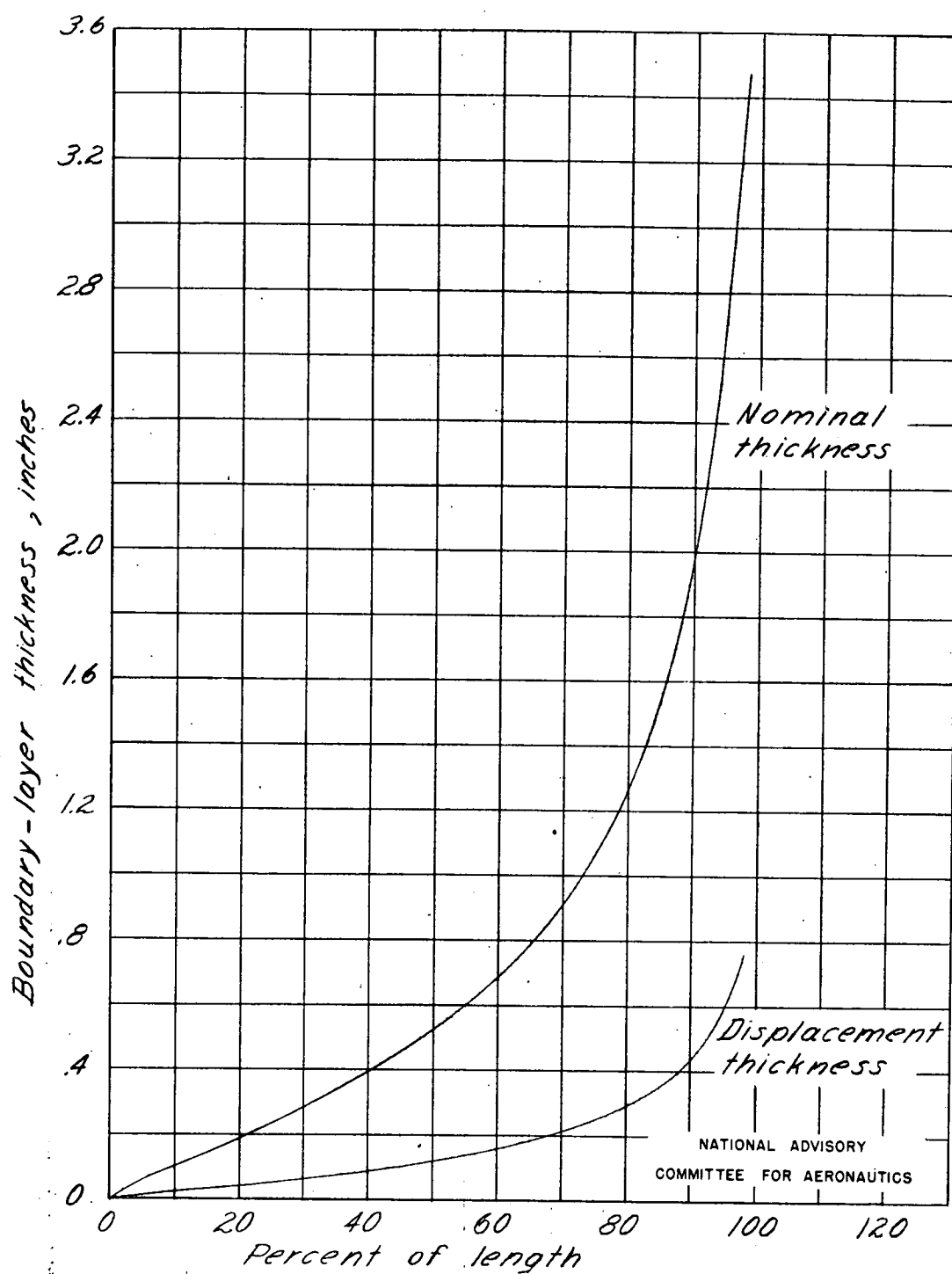
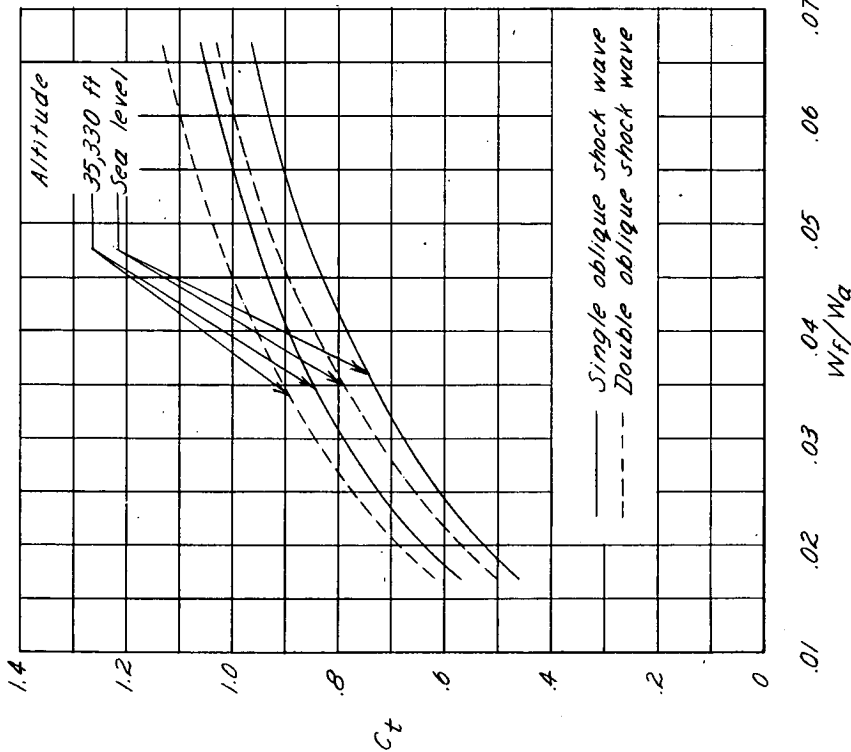
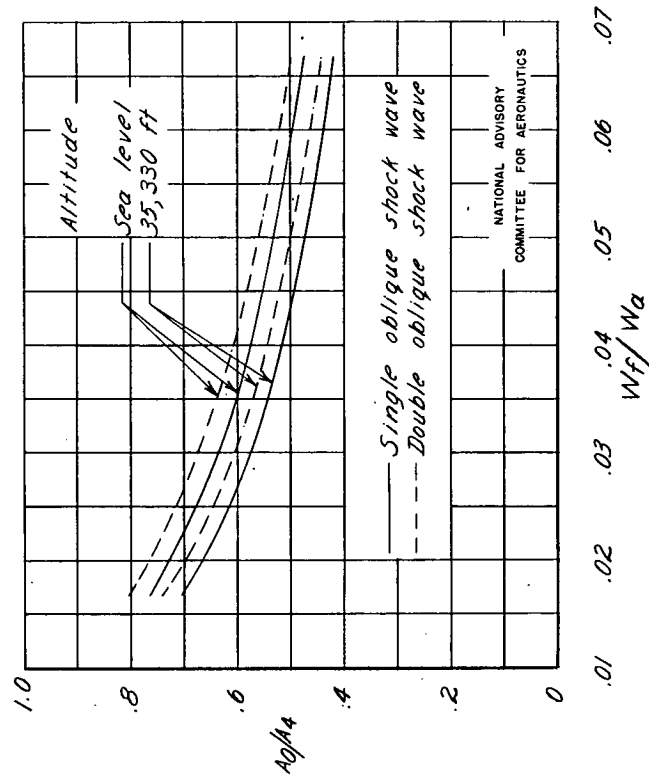


Figure 4.-Boundary-layer thickness on a parabolic body of revolution. Diameter, 10 inches; length, 100 inches; sea level; $M=2.0$.



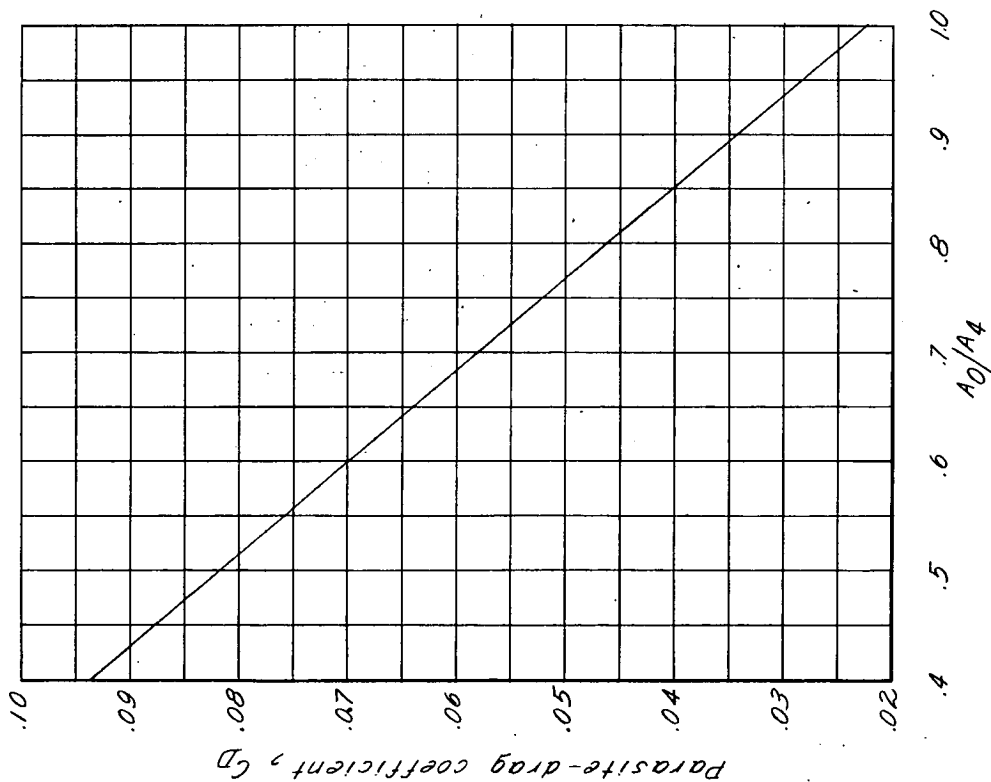
(a) Thrust coefficient based on projected frontal area of airfoil.

Figure 5.-Ducted-airfoil engine characteristics. $M=20$.



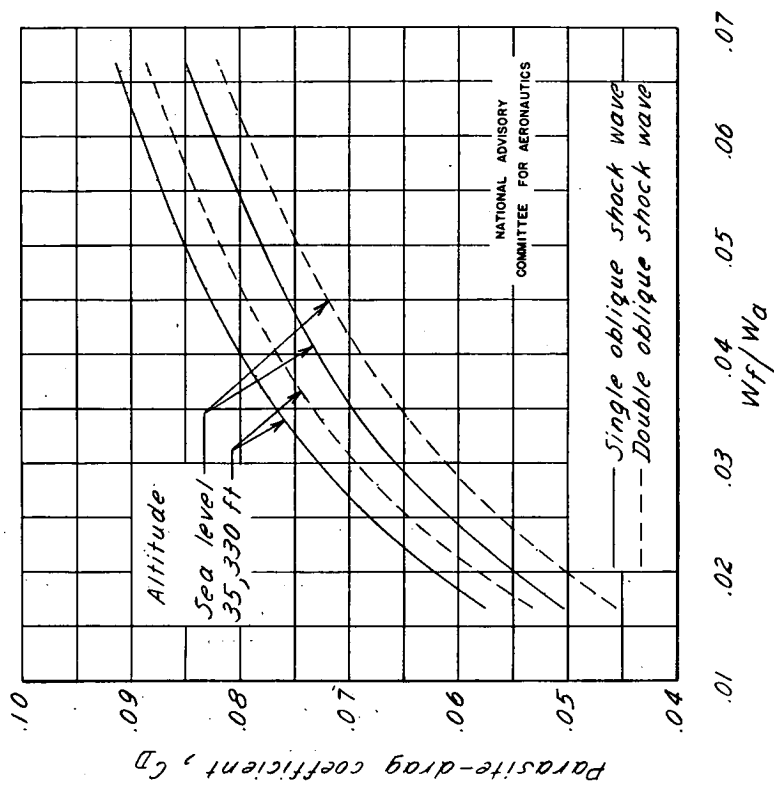
(b) Ratio of free-stream area of entering air to combustion-chamber area, A_0/A_4 .

Figure 5.- Concluded.



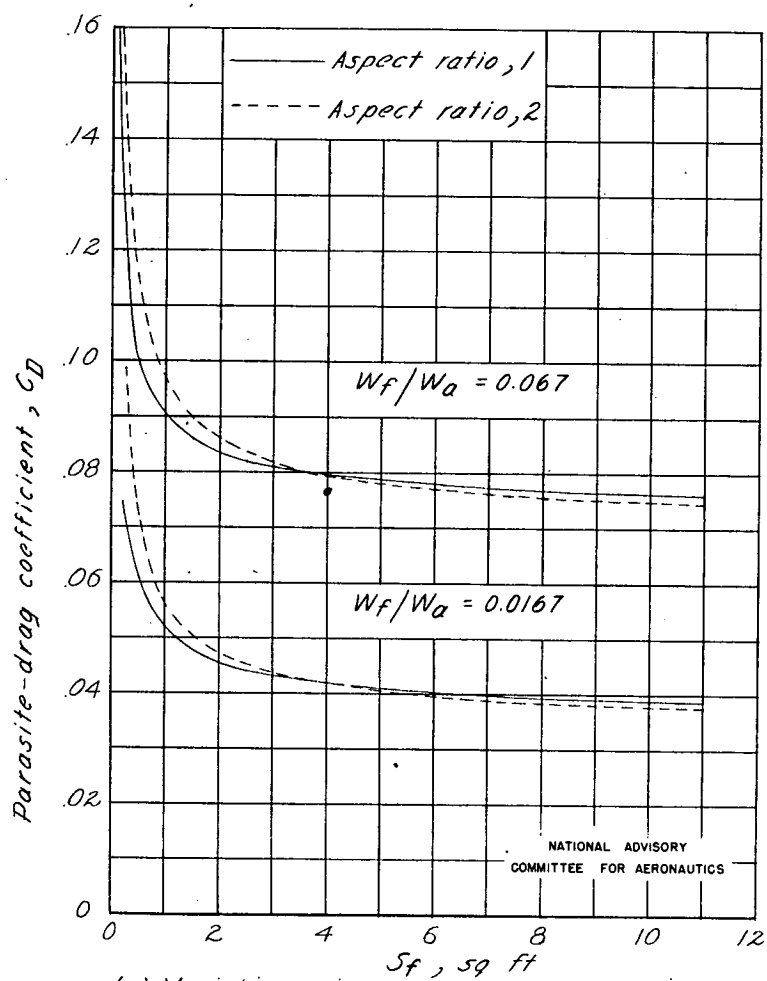
(a) Variation with A_0/A_4 .

Figure 6.-Ducted-airfoil parasite drag coefficient. $M=2.0$. Aspect ratio, 1.15.



(b) Variation with fuel-air ratio. Aspect ratio, 1.15.

Figure 6.-Continued:



(c) Variation with projected frontal area and aspect ratio.

Figure 6.- Concluded.

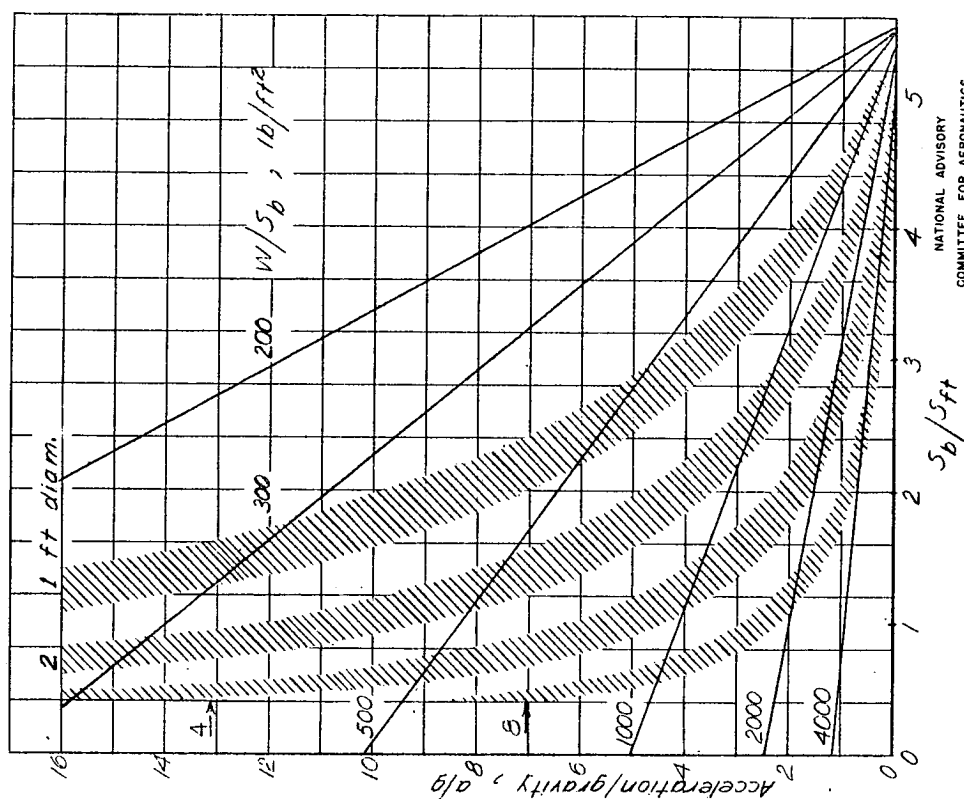


Figure 8.- Acceleration performance of ducted-tail ram jets. $W_f/W_0 = 0.067$; sea level; single oblique shock; $M = 2.0$; gross weight condition.

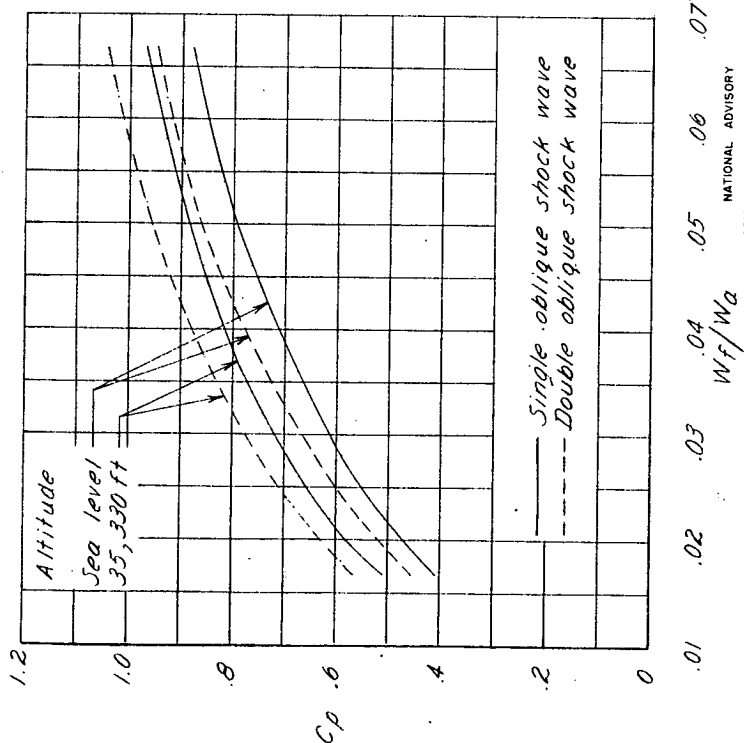


Figure 7.- Ducted-airfoil propulsive coefficients. $M = 2.0$.

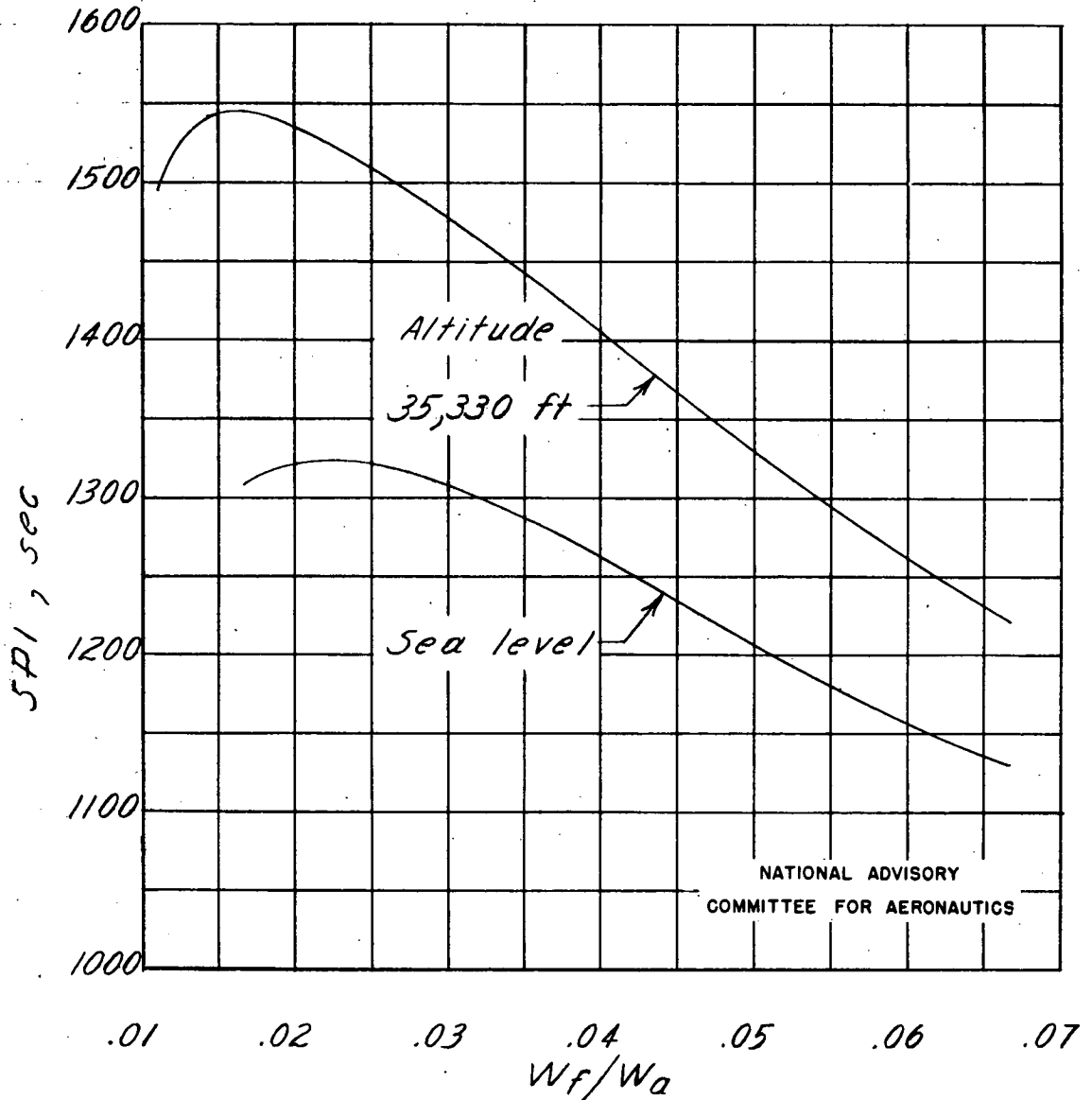


Figure 9.-Specific propulsive impulse. Single oblique shock wave. $M=2.0$.

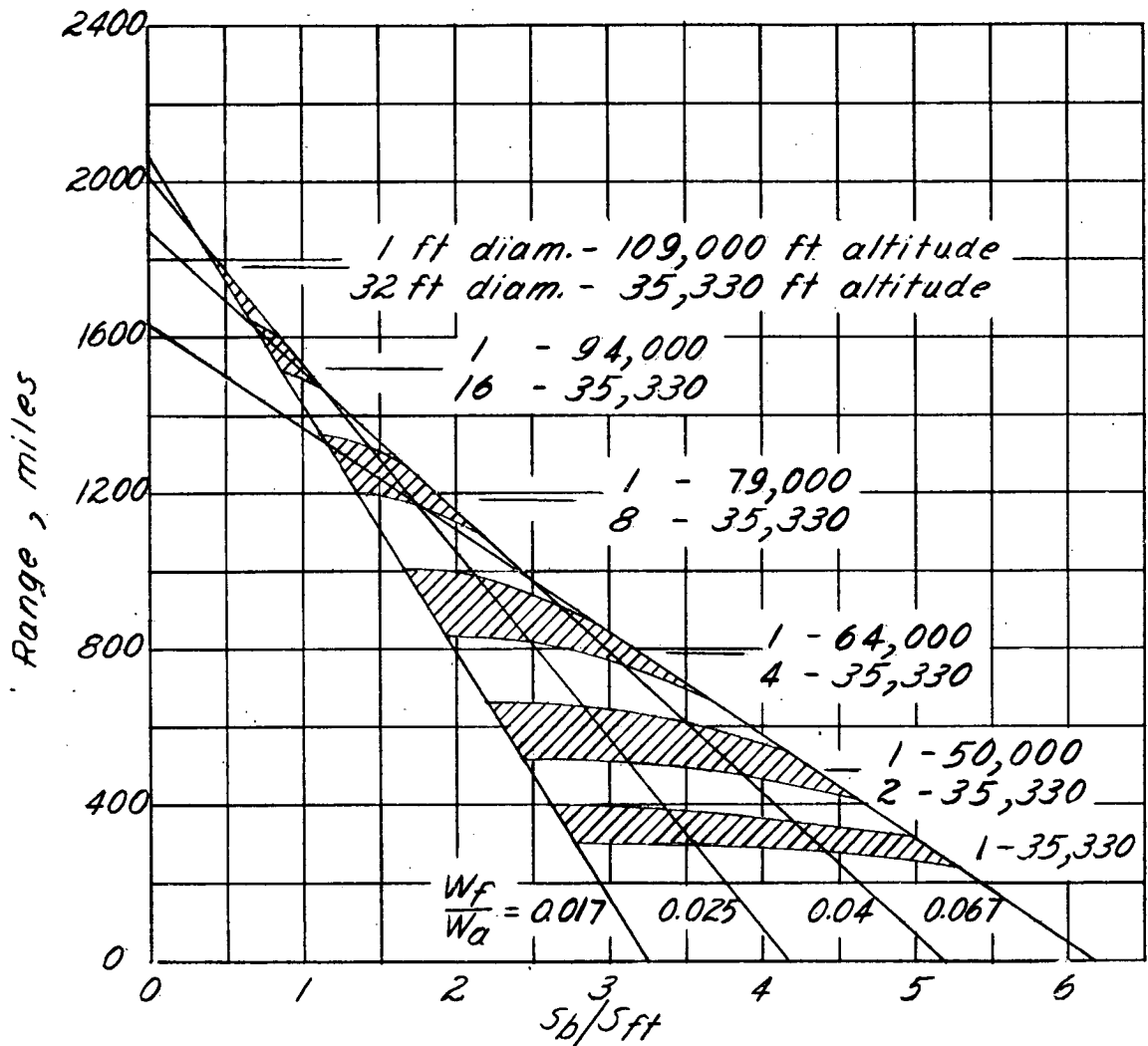


Figure 10. - Range chart for ducted-tail ram jets. Single oblique shock.

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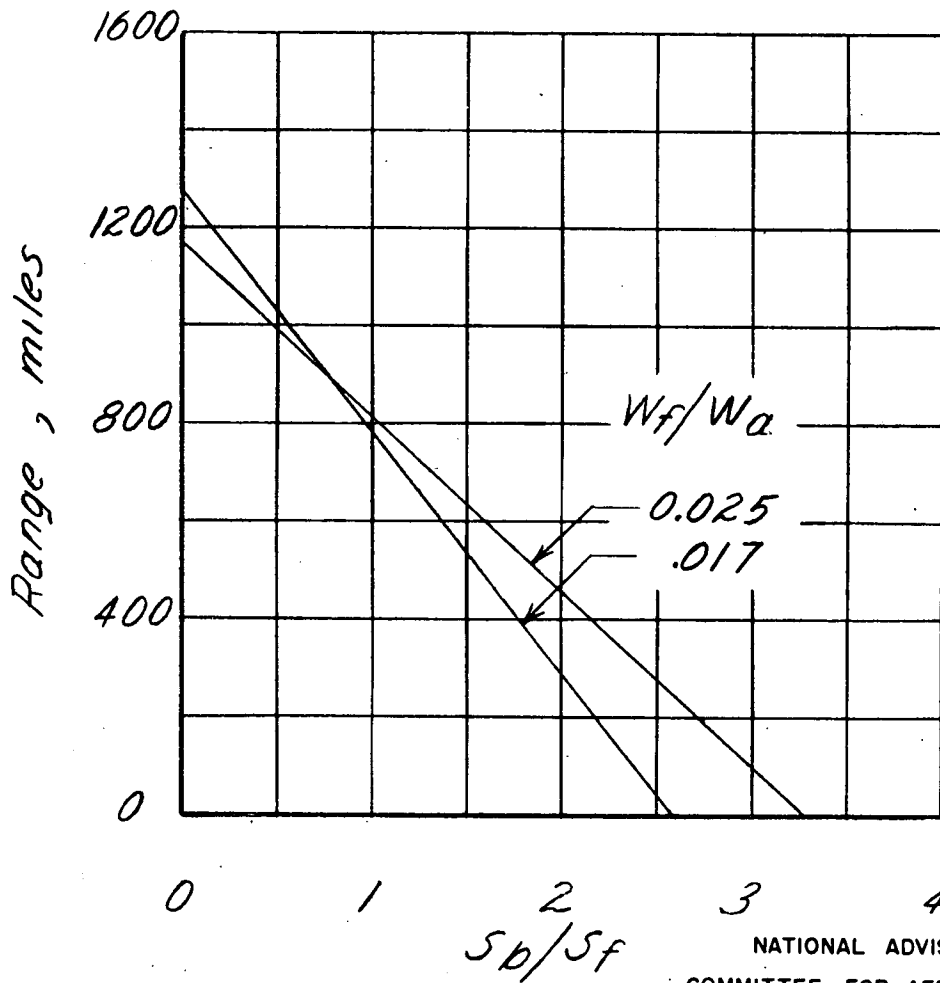


Figure 11.- Range chart for ducted wings.
Combustion chamber height/length = $1/2$.
Altitude, 35,330 ft. Single oblique shock.

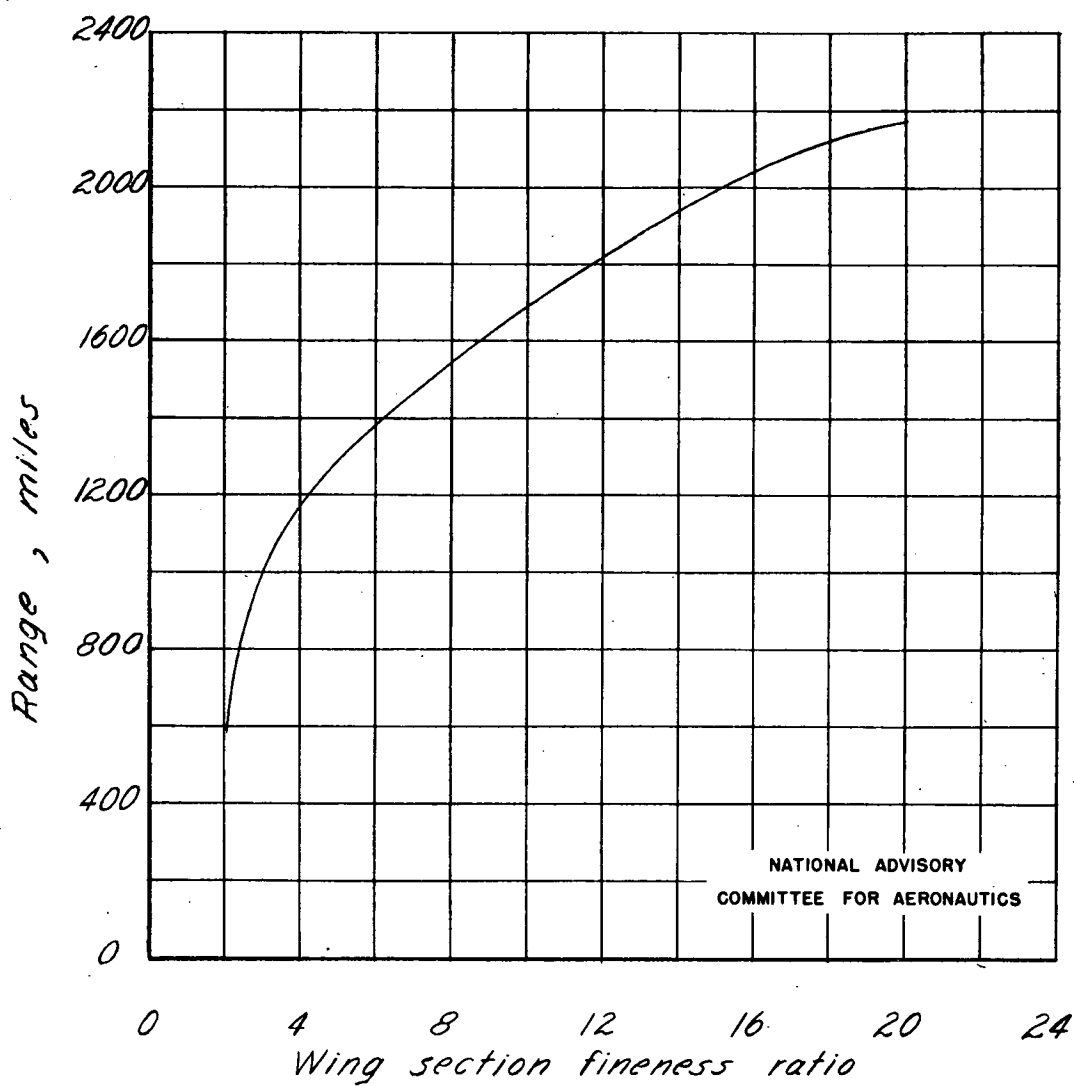


Figure 12.- Effect of ducted-wing fineness ratio on range. Altitude, 35,330 ft; wing aspect ratio, 1.15; single oblique shock wave. $S_b/S_f = 0$. $W_t/W_a = 0.017$.